AIAA 98-0147
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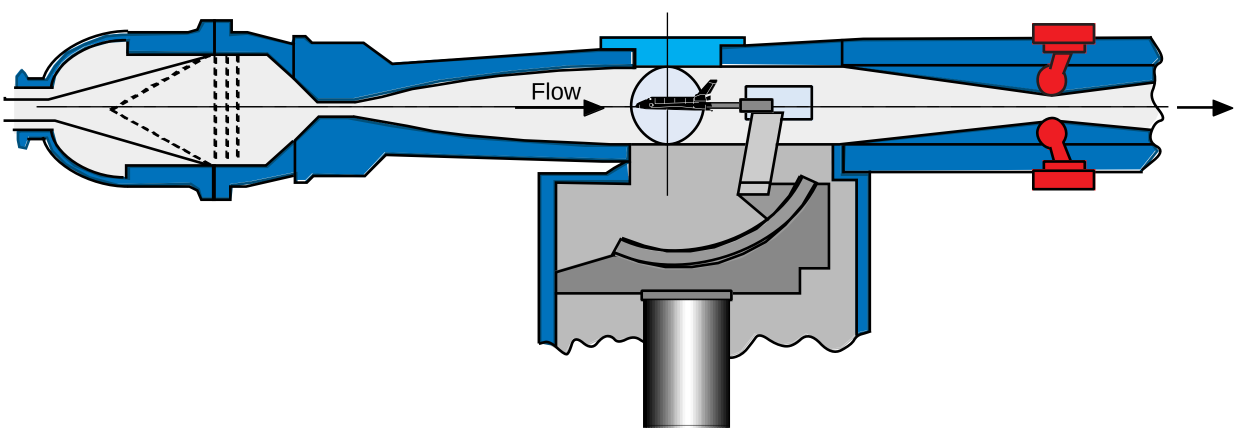

36th AIAA Aerospace Sciences Meeting & Exhibit
January 12-15, 1998 Reno, NV

Langley Aerothermodynamic Facilities Complex: Enhancements and Testing Capabilities

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Description, capabilities, recent upgrades, and utilization of the NASA Langley Research Center (LaRC) Aerothermodynamic Facilities Complex (AFC) are presented. The AFC consists of five hypersonic, blow-down-to-vacuum wind tunnels that collectively provide a range of Mach number from 6 to 20, unit Reynolds number from 0.04 to 22 million per foot and, most importantly for blunt configurations, normal shock density ratio from 4 to 12. These wide ranges of hypersonic simulation parameters are due, in part, to the use of three different test gases (air, helium, and tetrafluoromethane), thereby making several of the facilities unique. The Complex represents nearly three-fourths of the conventional (as opposed to impulse) -type hypersonic wind tunnels operational in this country. AFC facilities are used to assess and optimize the hypersonic aerodynamic performance and aeroheating characteristics of aerospace vehicle concepts and to provide benchmark aerodynamic/aeroheating data for generating the flight aerodynamic databook and final design of the thermal protection system (TPS) (e.g., establishment of flight limitations not to exceed TPS design limits). Modifications and enhancements of AFC hardware components and instrumentation have been pursued to increase capability, reliability, and productivity in support of programmatic goals. Examples illustrating facility utilization in recent years to generate essentially all of the experimental hypersonic aerodynamic and aeroheating information for high-priority, fast-paced Agency programs are presented. These programs include Phase I of the Reusable Launch Vehicle (RLV) Advanced Technology Demonstrator, X-33 program, Phase II of the X-33 program, X-34 program, the Hyper-X program (a Mach 5,7, and 10 airbreathing propulsion flight experiment), and the X-38 program (Experimental Crew Return Vehicle, X-CRV). Current upgrades/enhancements and future plans for the AFC are discussed.

Introduction

After nearly a decade of decline in hypersonics, a resurgence occurred in the mid-1980's, precipitated by planetary and earth-entry programs (e.g., Jones, 1987; Walberg, 1982; Naftel et al., 1989; Piland and Talay, 1989; Walberg, 1988). The primary impetus for this renewed interest was the National Aero-Space Plane (NASP), (e.g., Colladay, 1987), a single-stage-to-orbit concept employing airbreathing propulsion and representing a quantum leap in technologies. In the late 1980's and early 1990's, several studies were performed for various transatmospheric vehicles, single-stage-to-orbit (SSTO), reusable launch vehicle (RLV), and small-payload-to-orbit concepts. More recent studies include Phase I of the Reusable Launch Vehicle (RLV)/Advanced Technology Demonstrator, X-33 program, Phase II of the X-33 program (see Hamilton, H. H. III, et al., 1998), the X-34 program (e.g., Berry et al., 1998), the Hyper-X program (a Mach 5,7, and 10 airbreathing propulsion flight experiment), and the X-38 program (Experimental Crew Return Vehicle (X-CRV)) (Berry et al. 1997; Campbell et al. 1997; Loomis et al., 1997). A renewed interest in planetary rendezvous, atmospheric entry, and landing has occurred with the Mars Pathfinder Mission (Braun, et al., 1995; Gnoffo, et al., 1996; Horvath et al., 1996 and 1997; Gnoffo, et al., 1998) complemented by follow-on activities into the next century such as the Mars Microprobes (Mitcheltree et al., 1998). In response to the aerothermodynamic testing requirements for these and other programs, the NASA Langley

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Research Center, in particular the Aerothermodynamics Branch, has performed modifications, upgrades, and enhancements to the hypersonic facilities comprising the Aerothermodynamic Facilities Complex (AFC). Over a ten year span, facility upgrades have been advocated and accomplished via the minor Construction of Facilities (CoF) program and a single major CoF project occurring in 1989. In this same time period, significant upgrades to the instrumentation, signal conditioning, and data acquisition systems for these facilities were achieved via the Agency Aeronautical Wind Tunnel Revitalization Program and various other funding sources. The emphases of these upgrades were to provide improved flow quality, capability, productivity, and reliability.

The purpose of the paper is to: 1) provide an update on the AFC and discuss enhancements to the facilities and to measurement techniques that occurred since the publication of Micol 1995; 2) present AFC capabilities and facility-to-facility compatibility; 3) briefly discuss recent utilization of these facilities for several high priority fast-paced Agency programs; and 4) discuss future approved and proposed upgrades for the AFC.

Description, Capabilities, and Status of Facilities

Section Preface

The Langley Aerothermodynamic Facilities Complex (AFC) consists of five hypersonic, blowdown-to-vacuum, conventional (as opposed to impulse)-type wind tunnels that complement one another to provide wide ranges of free stream Mach number, M_∞, unit Reynolds number, R_∞, and normal shock density ratio, ρ_2/ρ_∞. Three different test gases are used as flow media: dry air, helium, and tetrafluoromethane (CF_4). The tunnels of the AFC are described to various degrees of detail in this section. A more complete description of the AFC facilities is presented in Micol, 1995 and Miller, 1992 and 1990, or by accessing the AFC website @ http://cyclops-mac.larc.nasa.gov/AFCwww /AFC.html.

Facility designations are defined using the following format. First, the size of the facility is given in terms of nozzle exit diameter or height, followed by its nominal Mach number and its test gas. When multiple facilities provide the same Mach number with the same test gas, then a pertinent feature such as high reservoir temperature is specified. For convenience, the operating conditions of the five facilities are summarized in Table 1 (see Hollis 1996).

All but three of the facilities are located in the Gas Dynamics Laboratory, building 1247. The 31-Inch Mach 10 Air and the 15-Inch Mach 6 High Temperature Air Tunnels are located in building 1251 (along with the Langley Unitary Plan Wind Tunnel) and the 20-Inch Mach 6 CF_4 Tunnel is located in building 1275. Photographs illustrating some basic features of these tunnels are shown in fig. 1.

20-Inch Mach 6 CF_4 Tunnel

The CF_4 tunnel is the only known operational, relatively low enthalpy, hypersonic wind tunnel in the country that generates a normal shock density ratio approaching that experienced during the hypervelocity portion of reentry into the Earth's atmosphere or during entry into some other planetary atmosphere. The significance of this capability to simulate the high density ratio aspect of a real (i.e., dissociated) gas such as occurs in flight has been illustrated for the AFE configuration as shown in fig. 2 (e.g., Micol, 1992) and the Space Shuttle Orbiter (see fig. 3; from Paulson and Brauckmann, 1995). The high density ratio and/or low ratio of specific heats (γ) aspect may be a significant factor in the hypersonic aerodynamics of moderately blunt to blunt bodies. (The density ratio produced in conventional-type hypersonic wind tunnels using air as the test gas is about 6, and only 4 in helium as compared to 12 in CF_4 at Mach 6; the advantages of using CF_4, which is three times heavier than air, as the test gas are discussed by Jones and Hunt, 1969).

The basic components of this tunnel include a CF_4 storage trailer, high-pressure CF_4 bottlefield, pressure regulator, salt-bath storage heater, a dual filtering system consisting of 20 and 5 micron in-line filters, settling chamber, nozzle, open-jet test section possessing a large number of optical accesses, diffuser, aftercooler, vacuum system, and CF_4 reclaimer. The CF_4 test gas is heated to a maximum temperature of approximately 1500°R by an electrical resistance heated storage bath-type heater containing a mixture of liquid heat-transfer salts (i.e., 50 percent Sodium Nitrate, $NaNO_3$ and 50 percent Potassium Nitrate, KNO_3). Flow is expanded through an axisymmetric contoured nozzle (Benton, 1989 and Benton et al., 1990) having a throat diameter of 0.446 in., an exit diameter of 20 in., and designed to produce Mach 6 at the exit. The settling chamber and nozzle were designed for a maximum pressure of 3000 psia and temperature of 1500°R.

Models are supported at the nozzle exit by a hydraulically-driven injection/ support mechanism for which the angle of attack may be varied from -10° to 50° and angle of sideslip from -5° to 5°. The injection time is variable from approximately 0.5 sec for heat-transfer tests to 2 sec for force and moment tests.

A more detailed description of this facility is presented by Midden and Miller, 1985 and Micol et al., 1992, along with a discussion of operational experience and calibration results. The CF_4 tunnel has been calibrated with the new nozzle (Micol, et al, 1992) for reservoir pressures from 100 psia to 2000 psia and temperatures from 1100°R to 1400°R, corresponding to a normal shock density ratio of 12 at Mach 6 and a range of unit free stream Reynolds numbers from 0.05 to 0.7 million per ft. Facility calibration results for this range of reservoir conditions is presented in fig. 4. As discussed in Micol, 1992, the level of flow uniformity achieved with the new nozzle is significantly improved when compared to the original nozzle. Pitot pressure is observed to vary $\pm$ 1.5 to $\pm$3.0 percent across the test core, depending on the reservoir pressure and temperature.

The facility received modifications in July 1997 via a F.Y. 1997 maintenance augmentation project. Modifications included a new replacement blower required to evacuate the tunnel test section prior to a tunnel run, a replacement cryogenic CF_4 pump (a critical component of the reclamation system), and a new 6000 gal. liquid nitrogen dewar also utilized in CF_4 reclamation.

20-Inch Mach 6 Tunnel

The Langley 20-Inch Mach 6 Tunnel, which became operational in 1958, is a blowdown wind tunnel that uses dry air as the test gas. Air from two high pressure bottlefields is transferred to a 600 psia reservoir and is heated within this reservoir to a maximum temperature of 1000°R by an electrical resistance heater. A double filtering system is employed having an upstream filter capable of capturing particles larger than 20 microns and a second filter rated at 5 microns. The filters are installed between the heater and settling chamber. The settling chamber contains a perforated conical baffle at the entrance and internal screens; the maximum operating pressure is 525 psia. A fixed-geometry, two-dimensional contoured nozzle is used; that is, the top and bottom walls of the nozzle are contoured and the sides are parallel. The nozzle throat is 0.399 in. by 20 in., the test section is 20.5 in. by 20 in., and the nozzle length from the throat to the test section window center is 7.45 ft. This tunnel is equipped with an adjustable second minimum and exhausts either into combined 41 ft diameter and 60 ft diameter vacuum spheres, a 100 ft diameter vacuum sphere, or to the atmosphere through an annular air ejector. The maximum run time is 2 minutes with the two spheres, 10-15 minutes with the single large sphere, and 20 minutes with the ejector.

Models are mounted on the injection system located in a housing below the closed test section. This system includes a computer operated sting support system capable of moving the model through an angle of attack range of -5° to +55° for angles of sideslip of 0° to $\pm$10°. Injection time of the model for heat-transfer tests can be as rapid as 0.5 sec. For force and moment tests, the injection time is adjusted to 1.2 sec with a maximum acceleration of 2 g.

A description and calibration of the 20-Inch Mach 6 Tunnel are presented by Goldberg and Hefner, 1971, and calibration results are presented by Miller and Gnoffo, 1981. More recently, a facility calibration was performed for a range of reservoir pressures from 30 psia to 500 psia and temperatures from 760°R to nearly 1000°R, corresponding to free stream Mach numbers from 5.8 to 6.1 and unit Reynolds numbers from 0.5 to 9 million per ft. As shown in fig. 5, pitot-pressure surveys measured over this reservoir pressure range are reasonably flat, symmetrical about the nozzle centerline, and do not indicate the presence of any severe flow disturbances. Pitot-pressure, hence dynamic pressure variations across the test core, is $\pm$ 2.2 percent for a reservoir pressure of 30 psia and less than $\pm$ 1.0 percent for reservoir pressures in excess of 125 psia. Core size varies from 12 to 14 in. as freestream Reynolds number increases.

The normal shock density ratio for this facility is 5.3 and ideal gas behavior is achieved within the entire flow sequence (i.e., $\gamma = 1.4$ everywhere); thus, models may be tested in this facility and the 20-Inch Mach 6 CF_4 Tunnel at the same values of free stream Mach number and unit Reynolds number to determine the effect of a variation in density ratio (see figs. 2 and 3). Similarly, models may be tested in this facility and the 31-Inch Mach 10 Tunnel (to be discussed subsequently) to examine compressibility (Mach number) effects. Also, this facility presents the opportunity to examine boundary layer transition to turbulence on candidate aerospace vehicle concepts since the Reynolds number may be varied by a factor of 16 (see, for example, fig. 6 where Reynolds number is varied by a factor of 8).

15-Inch Mach 6 High Temperature Tunnel

The basic components of this tunnel include an 865 ft^3 bottlefield; a 5 - MW AC resistance heater consisting of 216 electrically-heated tubes through which the air flows, and mounted vertically to accommodate thermal expansion; 5 micron in-line filter; pressure regulator; settling chamber; an axisymmetric contoured nozzle having a throat diameter of 1.81 in.; nozzle exit diameter of 14.57 in., and length of 75.6 in.; a walk-in open-jet test section identical to the 20-Inch Mach 6 CF_4 Tunnel and with numerous optical accesses; a hydraulically-driven injection/retraction support mechanism for which the angle of attack may be varied from -10° to 50° and sideslip between ±10°; variable area diffuser; an aftercooler; and a vacuum system shared with the 31-Inch Mach 10 Tunnel. The electrical heater is capable of heating the air to 1500°R at a maximum pressure of 500 psia.

As discussed by Hodge, 1992, the original Mach 10 nozzle for this facility was converted to Mach 6 via a new nozzle design. The new Mach 6 nozzle was fabricated and installed in the late 1991/early 1992 timeframe. Calibration results (fig. 7) obtained via pitot-pressure surveys for this new Mach 6 nozzle revealed a high level of flow uniformity both radially and axially and a test core free of centerline disturbances typical of hypersonic axisymmetric contoured nozzles. The run time available with the much larger mass flow Mach 6 nozzle, as compared to the original Mach 10 nozzle, is about 90 sec.

Modifications and upgrades to this tunnel since the report by Micol, 1995, include construction of a new variable area diffuser and a heat exchanger. These two items were added under a single Construction of Facilities (CofF) project in order to improve the operational envelope of the facility for all classes of models (i.e., slender, moderately blunt, blunt) and to replace the antiquated and unreliable tube sheet heat exchanger with an improved cooling capacity finned-type heat exchanger.

A significant improvement to the operational efficiency, hence productivity of both the 15-Inch Mach 6 HTT and the 31-Inch Mach 10 Air facilities, was achieved by the addition of a three stage steam ejector in April 1997. This additional pumping capability reduced pumpdown times between runs by one-third (i.e., from 1.5 hours to approximately 40 minutes to evacuate two 41 ft spheres from approximately 200 mm Hg to 1 mm Hg), thereby increasing the frequency at which these facilities can operate (i.e., the number of wind tunnel runs per 8 hour shift was increased by approximately 2).

31-Inch Mach 10 Tunnel

The 31-Inch Mach 10 Tunnel operates in the blowdown mode (though designed as a blowdown start, continuous running facility). It consists of a high pressure air storage system having a volume of $865 ft^3$ and rated at 4400 psia maximum, a 12.5-MW electrical resistance heater located in a vertical pressure vessel, a 5-micron in-line filter, settling chamber, three dimensional contoured nozzle, 31 Inch square closed (as opposed to open jet) test section, adjustable second minimum, aftercooler, vacuum spheres and vacuum pumps. The settling chamber, nozzle throat section, test section, adjustable second minimum, and subsonic diffuser are all water cooled. The 12-in. diameter settling chamber is equipped with screens at the upstream end and is faired into the upstream end of a square cross section nozzle having a 1.07 in. square throat. The throat section, modified in July 1992 to replace the original segmented throat, is one piece fabricated from beryllium copper, and backside water-cooled with the cooling water system operating at 500 psia. The surface of this nozzle is uncoated. (The original nozzle throat section surface was coated with zirconium and required periodic refurbishment.)

Models are supported on a hydraulically-operated, sidewall-mounted injection system. Models can be injected to the nozzle centerline in less than 0.6 sec and the angle of attack varied from -10^o to $+45^o$ with a straight sting (i.e., being constrained to within a $\pm$ 7 in. inviscid core where pitot pressure distribution varies approximately ± 1.0 percent). Sideslip range is $\pm 5^o$ and both angle of attack and yaw are remotely controlled by a computer. With the second minimum closed to about 25 percent of the maximum test section cross section area and the use of two 41-ft-diameter vacuum spheres, run times of approximately 60 sec are achieved. The recent connection of the vacuum system to a 60-ft vacuum sphere (commonly referred to as the 60 foot Space Simulator) increased run times of this facility and the 15-Inch Mach 6 HTT by a factor of 2. As a result of a FY 1996 Minor CofF project, this facility (and naturally, the 15 Inch Mach 6 HTT) has benefitted from the addition of a three stage steam ejector as discussed previously. This upgrade has provided a significant increase in pumping capacity, thereby reducing turn-around time and increasing productivity by 1.5 to 2 times. The steam ejector has allowed this facility to approach the production-like characteristic of the 20-Inch Mach 6 Tunnel.

Calibration results using the one-piece nozzle throat section are shown in fig. 8. This facility has been calibrated for reservoir pressures from 350 psia to 1450 psia for a temperature of approximately 1800^oR, corresponding to a range of free-stream unit Reynolds number of 0.50 to 2.1 million per ft. The flow in the center region, provided by the water cooled 3-D nozzle, is quite uniform at all values of reservoir pressure (see fig. 8; the variation in pitot pressure across the center 7in. of the test core generally is within the uncertainty of the measurement, i.e., less than ± 1 percent). The test core varies from about 12 in. by 12 in. at the lowest Reynolds number condition to approximately 14 in. by 14. in at the highest value.

22-Inch Mach 15/20 Helium Tunnel

This helium tunnel is an intermittent, closed-cycle, blow-down tunnel that uses high purity helium (less than 40 parts per million impurities at purchase) as the test gas. The components of the tunnel circuit include a high pressure storage system designed for 5000 psia, an in-line electrical resistance heater capable of heating the gas to a maximum temperature of 1100^oR, a 5 micron in-line filter, and a settling chamber. The flow is expanded through a 0.622-in. diameter throat and an axisymmetric contoured nozzle, designed to provide Mach 20 flow at the exit, into a contoured test section having a length of 11.6 ft and a maximum diameter of 22-in. (This facility also has an interchangeable second nozzle designed for

Mach 15 having a throat diameter of 1.23 in.) The test section is equipped with a hydraulically driven model injection system. The flow is decelerated by means of a two-dimensional, constant-area diffuser before entering two interconnected 60-ft-diameter vacuum spheres. Typical run time is 30 sec, but run times of 60 sec have been achieved using two additional 60-ft-diameter spheres (those previously connected to the 60-Inch Mach 18 Helium Tunnel which is now being dismantled). The helium collected in the spheres is recompressed to 5000 psia and is passed through molecular sieve filter beds, LN_2 cold traps, and a purifier using silica-gel dryer maintained at 140^oR to reduce the contaminating agents to less than 0.02 percent by volume. After purification, the helium is stored in high pressure bottles for reuse.

This facility, like the CF_4 tunnel, provides a test capability that does not exist elsewhere in this country. This helium tunnel is the only high hypersonic Mach number facility for which the free stream flow and flow within the shock layer of a model are everywhere thermally and calorically perfect. The helium test gas does not require heating for Mach numbers less than about 28 because helium may be expanded to temperatures as low as 3^o to 4^oR without liquefying at the corresponding low pressures. The very low free stream temperature at Mach 20 for ambient reservoir temperature corresponds to relatively high values of free stream Reynolds number. (Recent calibrations at a reservoir pressure of 3200 psia provided $Re_\infty = 22 \times 10^6$/ft at Mach 20.) The resulting flow conditions often simulate entry flight Mach/Reynolds numbers, hence viscous interaction parameters, for a wide variety of entry vehicles. Calibration results for the current axisymmetric contoured Mach 20 helium nozzle for a range of reservoir pressure from 500 to 1500 psia and ambient temperature corresponding to a variation in Reynolds number from 4.0 to 11.3 x 10^6 ft^{-1} is presented in fig. 9. (For a historical perspective on the development of this nozzle and the Mach 15 nozzle, see Micol, 1995.) Calibration results for the Mach 15 nozzle at ambient conditions are presented in fig. 10 for a range of reservoir pressures from 500 to 1700 psia corresponding to a variation in R_∞ from 4.8 to 15.8 x 10^6 ft^{-1}.

Synergism of Facilities

As mentioned previously, examination of a wide array of simulation parameters is made possible via the synergism between facilities within the AFC. The following example of a test matrix for a slender body, whereby all four basic hypersonic simulation parameters may be important (as opposed to a blunt body, where only one parameter may be significant at hypersonic, continuum conditions) illustrates that collectively the AFC tunnels provide a unique national capability.

To measure effect of	For given	Test in	and
M_∞	R_∞, γ_∞, Tw/Taw	15" M6 HTA	31" M10
M_∞	R_∞, γ_∞, Tw/Taw	22" M15 He	22" M20 He
R_∞	M_∞, γ_∞ Tw/Taw	All facilities	
γ_∞	M_∞, R_∞, Tw/Taw	15" M6 HTA	20" M6 CF_4
Tw/ Taw	M_∞, R_∞, γ_∞	15" M6 HTA	
Tw/ Taw	M_∞, R_∞, γ_∞	22" M20 He	(with and without heater)

Examples of the synergism provided by the AFC to simulate hypersonic flight conditions for proposed aerospace vehicles are shown in figs. 2, 3, and 6.

Instrumentation and Testing Techniques

Section Preface

Instrumentation and measurement techniques routinely utilized in the tunnels of the AFC are discussed in this section. (A more detailed description of each measurement technique is found in Micol, 1995.) As a result of the utilization of these small research-type facilities in a production-type mode in the last few years, approximately 60 percent of the testing has been devoted to force and moment and 40 percent to aeroheating. (Very few tests have been devoted to pressure.)

The most significant change in recent years in aerothermodynamic measurements at Langley has been the rapidly growing application of a relative intensity (two-color), thermographic phosphor technique developed by G. M. Buck (Buck, 1988, 1991). This technique has evolved from initially obtaining qualitative measurements of surface temperature distributions (thermal mappings) to the present capability of obtaining global quantitative heat-transfer measurements (Merski, 1991, 1998). It allows for fast-paced aerothermodynamic studies since the model, once coated with phosphors (in contrast to other thermal coatings such as phase change paint), can be reused indefinitely. The associated video recording system and sophisticated software provide thermal mapping displays immediately after a run. An overview discussing the history, evolution, and improvements to this technique is presented in Merski, 1998. This technique provides a means of acquiring complementary aeroheating databases as configuration aerodynamic databases are being developed.

Forces and Moments

A comprehensive inventory of internal strain gage balances is maintained at Langley. Balances are sting or strut supported, and those used most frequently in the tunnels of the AFC have an outside diameter of 0.56 and cover a wide range of maximum design loads applicable to blunt, high drag models and to slender, high lift models. These water cooled balances are 6 components and require an excitation voltage of 5 volts. The general procedure used in most tunnels is to calibrate all 6 components, or as a minimum, the axial- and normal-force and pitching moment components, of the balance by loading the balance with weights and recording the output. These data are compared to the formal calibration data prior to the installation of the model to assure the balance is operating within acceptable tolerances. A tare run with the model mounted to the balance is performed over the desired angle of attack range prior to the test series. Straight and bent stings and blade mounted struts of various shapes and sizes are available to allow testing over a wide range of angle of attack. In preparation for Phase I and II of X-33 and X-34 programs, eight new strain gage balances were fabricated bringing the total number of water cooled balances available for use in the AFC to 32.

Pressure

Base and cavity pressures are routinely measured with electronically scanned pressure (ESP) piezoresistive (silicon) sensors. ESP modules are available with different pressure ranges and typically contain 16, 32, 48, or 64 scanners; these modules may be located within or in close proximity to the model. All facilities within the AFC have Pressure Systems, Inc. (PSI) model 8400 measurement systems and presently 512 channels of pressure measurements are available in these facilities. High volume, multirange, variable capacitance diaphragm-type transducers are available for certain applications (e.g., very low pressure measurements like wall static pressures and test section or box static pressure). Most facilities have 10 to 20 variable-capacitance transducers.

Heat Transfer

Thin-Skin and Thin-Film Gages

Conventional thin-skin transient calorimeter technique and/or the thin-film resistance thermometer technique are generally used to obtain benchmark heating studies within the AFC. The latter technique uses thin-film gages originally developed for use in impulse-type facilities (Miller, 1984; Miller et al., 1985). The routine testing of models having a large number of mechanically deposited (i.e., sputtered, vapor deposited) thin-film heat-transfer gages in conventional-type hypersonic wind tunnels was pioneered at Langley in the early 1980's and adopted as the "standard" for aeroheating studies in the AFC until the arrival of the phosphor thermography technique to be discussed subsequently. As a result of the revitalization program, each facility is capable of acquiring 128 channels of thin-skin/ thermocouple-type data and 128 channels of thin-film data. The 20 Inch Mach 6 and the 31 Inch Mach 10 Tunnels were upgraded to 256 channels for both techniques.

Previously data reduction of heat-transfer rates for the thin-film technique was accomplished using the measured temperature-time histories and the numerical method of Cook, 1966, or method of Kendall, et al., 1967. More recently a new data reduction scheme has been developed at Langley (Hollis, 1995). This new procedure is known as the One-Dimensional Hypersonic Experimental Aero-Thermodynamic data reduction code or 1DHEAT. 1DHEAT is a fast user-friendly scheme for heat-transfer data reduction which employs both analytical and finite volume models. These models account for variable thermal properties and can be used to reduce either thin-film or coaxial thermocouple data. The code also includes the ability to deal with multiple layer substrates using the finite-volume model.

<u>**Thermographic Phosphors**</u>.- Historical detail of the development of the two-color thermographic phosphor technique (Buck, 1988, 1991; Merski, 1991) is presented in Miller, 1990 and 1992, and Micol, 1995. The present description of the phosphor thermography system outlines advances made since 1995. An overview of this technique is presented in Merski, 1998.

The rapid advances in image processing technology which have occurred in recent years have made digital optical measurement techniques practical in the wind tunnel. One such optical acquisition method (see fig. 11) is the two-color relative-intensity phosphor thermography technique, which is currently being applied to aerothermodynamic testing in Langley's hypersonic wind tunnels. With this technique, ceramic wind tunnel models are fabricated and coated with phosphors which fluoresce in two regions of the visible spectrum when illuminated with ultraviolet light. The fluorescence intensity is dependent upon the amount of incident ultraviolet light and the temperature to which the phosphor is exposed. Fluorescence intensity images of an illuminated phosphor model exposed to a hypersonic stream are acquired by a color video camera. Surface temperature mappings can be calculated for portions of the model within the field of view of the camera. This is done by utilizing the green and red camera outputs. The resulting intensity images are converted to temperature mappings via a temperature-intensity calibration of the phosphor coating. Currently, this calibration is valid over a temperature range from 22 (532 °R) to 170 (800 °R) degrees Celsius. Heat-transfer rates may then be calculated at every point on the image (and hence globally on the model) from time-sequenced images taken during the wind tunnel run. (One-dimensional heat conduction into a semi-infinite slab is assumed in the heat-transfer rate calculation.)

Data acquisition is performed with PC-based video acquisition systems and color solid-state video 3-CCD (Charge Coupled Device) cameras. These systems digitize phosphor fluorescence intensity images to resolutions of 512 x 481 pixels which are then transferred to UNIX workstations for data reduction. To analyze the large amount of data acquired with phosphor thermography, Langley has developed a workstation-based image processing package called IHEAT. IHEAT is written in a user-friendly windowing format and consists of six programs to handle system calibrations along with data reduction, editing, and viewing. Data can be reduced to global heat transfer images within minutes following a run using IHEAT. An automated routine also provides plots of heating along axial and longitudinal cuts.

A new methodology known as EXTRAP is introduced in Merski, 1998, whereby global phosphor thermography wind tunnel data can be extrapolated to flight heating levels. Given trajectory information and IHEAT images, EXTRAP computes extrapolated global flight surface temperatures and heat transfer rates. In Merski, 1998, code features are presented and calculated surface temperatures and heat transfer rates for the X-34 flight vehicle using a Navier-Stokes solver known as Laura (Langley Aerothermodynamic Upwind Relaxation

Algorithm) are compared to extrapolated wind tunnel data. Comparisons of calculated surface temperature and heat transfer rates to extrapolated data are in good agreement.

Fabrication of wind tunnel models for phosphor testing is a critical component in the technique. In order to obtain accurate heat transfer data using the one-dimensional heat conduction equation, models need to be made of a material with a low thermal diffusivity and well-defined, uniform, isotropic thermal properties. Also, the models must be durable for repeated use in the wind tunnel and they should be subject to minimal deformation when thermally cycled. To meet these requirements, Langley has developed a unique, silica ceramic slip casting method. Patterns for the models are typically made using a numerical cutting machine or with the stereo-lithography process. Using these patterns, investment molds are created. Ceramic slip is poured into the molds and after the ceramic hardens, the investment molds are crumbled off leaving ceramic shells. These shells are next fired in a kiln and potted with support hardware (e.g., stings or blade-mounts). The slip casting method provides the means to capture relatively fine configuration details and thin ceramic sections such as wings and fins. Model lengths are typically 10 to 14 inches; however, models up to 30 inches in length have been fabricated for testing at low angles of attack. In addition, the slip-casting method is a rapid process whereby, in three to four weeks, a full array of inexpensive models can be fabricated, complete with various perturbations needed for a configuration build-up scheme such as variable nose radii and control flap deflections.

Once the models are fabricated, they must be coated with phosphor crystals. Currently, the phosphors are suspended in a silica ceramic binder and applied with an air brush. Final coating thicknesses of approximately 0.001 inches have been measured. The coating method which has been developed, produces robust coatings which do not require re-application between runs, thereby significantly enhancing the efficiency of the phosphor technique.

The thermographic phosphor technique offers several advantages over discrete gage techniques. The most significant advantage is that surface temperatures are determined in a global sense; that is, within the pixel resolution of the camera and useful range of the phosphor, surface temperatures are obtained at every point on the model surface within view of the camera and at various times during the run. As mentioned previously, phosphor models do not require cleaning and recoating between runs, thereby allowing for more efficient operation of the facility. The thermographic phosphor system provides heating distributions on models for analysis within minutes after a run. This technique has replaced the phase change paint technique and is rapidly advancing to a quantitative level sufficient to replace the thin-film gage technique.

Infrared Emission.- The surface temperature of a model may be inferred from observations of its radiation at infrared wavelengths. Infrared imaging has been used in hypersonic wind tunnels for over two decades (e.g., Schultz and Jones, 1973; Neumann, 1988). The infrared technique received renewed interest within the hypersonic testing community in the early 1990's (Daryabeigi, 1992) because of recent advances providing higher image resolution, standard video format data acquisition, radiometric (absolute temperature) measurements and computer based digitized image processors. A commercially built infrared imaging system (Inframetrics 600; liquid nitrogen cooled) is available for use in the Langley AFC; it is a single detector imager employing optical-mechanical scanning to provide two-dimensional temperature distributions for the 8 to 12 μm band pass of the electromagnetic spectrum. The system has a frame frequency of 30 Hz providing 380 samples per line and 200 lines per frame. Standard video format is used. Stored information is manipulated and then transferred to a digital image processor. The precision of these infrared imaging systems for providing steady-state temperature measurements has been evaluated in the laboratory for temperatures from $400^{o}R$ to $1390^{o}R$ and found to be within 3 percent (Daryabeigi, 1992). Wind tunnel test sections have been equipped with zinc selenide windows, having a suitable anti-reflection coating, that are capable of transmitting 8 to 12 μm wavelengths. Models made of insulating material are preferred because of inherent higher emissivity in the infrared spectrum which provides the camera with a stronger signal. Presently, the surface emissivity is measured as a function of wavelength at ambient temperatures only; future plans call for the measurement of emissivity over a range of temperatures. The overall sensitivity depends on the viewing angle. (For a dielectric material such as fused silica ceramic there is no significant reduction in emissivity until the viewing angle exceeds 60° (Rhode, 1997).) In his paper, Daryabeigi addresses the application of the infrared technique in the AFC.

A second infrared imaging system available within the AFC is an Inframetrics 760 that has a stirling cycle cooler, 8-12 μm spectral response, 30 Hz framing rate with 380 samples per line and 200 lines per frame, 40° horizontal by 30° vertical field of view (FOV) and 12-bit digitizer. This system is utilized primarily in the Mach 6 and 10 air tunnels in which a major portion of aerothermodynamic testing at Langley is performed.

Within the AFC this technique has been used to detect complex heating patterns, including boundary layer transition to turbulence, on force and moment, pressure and heat transfer models, and to examine plumes doped with freon. It should be noted that the infrared technique is not applicable to the 20-Inch Mach 6 CF_4 tunnel because of emission absorption by this heavy gas.

Flow Visualization.- Included in this category are schlieren systems, vapor screens, electron-beam flow field visualization, and oil flow. Although a holographic interferometry system was used at the 20-Inch Mach 6 CF_4 Tunnel for several years (Burner and Midden, 1977), no such systems are incorporated into the AFC at present.

All of the AFC tunnels were originally equipped with pulsed white-light, Z-pattern, single-pass schlieren systems (e.g., Compton and Spencer, 1967). Of the active tunnels, only the 31-Inch Mach 10 and 22-Inch Mach 20 He Tunnels do not routinely use schlieren systems.

Schlieren quality for the 22-Inch Mach 20 Helium Tunnel is poor because of the low density gradients at the high Mach numbers of this facility. Alternate flow visualization techniques were sought and the facility was equipped with an electron gun designed and fabricated at LaRC (e.g., Harvey and Hunter, 1975; Woods and Arrington, 1972; Honaker et al., 1979). The electron gun was designed to induce fluorescence that could be spectrally analyzed to infer density and temperature nonintrusively. The advantages of induced fluorescent flow visualization were recognized during the early attempts to measure density and temperature. Since the shock structure and characteristics of the flow field were revealed by the electron beam, the technique was adopted for flow visualization. At present, shakedown of a new commercially available electron gun (from Kimball Physics) is underway.

Vapor screen is a technique that provides a cross section view of the shock and boundary layer about a model, and is particularly useful for observing vortices. For this technique, the reservoir temperature is reduced sufficiently or the dew point raised to cause flow condensation at the test section. Light is passed through a slit to produce a screen effect or sheet of light across the flow. Flow details about a model inserted into this sheet can be photographed. This technique is sometimes used in the 20-Inch Mach 6 and 31-Inch Mach 10 tunnels, but must be viewed as purely qualitative. It also should be viewed with caution because of the adverse effects lowering the temperature has on flow uniformity and changes in flow conditions.

A widely used technique to provide information on model surface streamline patterns is oil-flow. Usually, models are painted black, sprayed with a mixture of oils of various viscosities mixed with white artist pigment, and then injected rapidly into the flow. Movement of the oil is photographed while the model is in the flow. In several facilities, insignificant movement of oil on the surface is experienced during the retraction process; hence, the model can be removed from the tunnel after the run and the flow patterns photographed, generally with a Kodak DCS 460 digital camera, in detail over the entire surface. Oil-flow tests are generally performed on each model to augment interpretation of force and moment, pressure, and thermal mapping/heat transfer measurements.

Flow Field Surveys.- Techniques used to measure model flow field properties at various locations within the shock layer, including boundary and shear layers, employ probes which are likely to perturb the flow; that is, they are intrusive. A large amount of effort was devoted to the development of nonintrusive or optical flow field measurement techniques in the AFC in the late 1960's and the 1970's; however, there is minimum application of these technologies today. Very little flow field survey work with mechanical devices such as hot wire (e.g., Wagner and Weinstein, 1974), hot-films, total pressure, and static pressure probes has been performed in recent years. This is primarily due to programmatic changes; that is, the tremendous demands on facilities of the AFC to support high-priority, focused Agency programs such as X-33 and X-34.

The capability for survey studies exists and centers on the application of miniature (0.013 in. diameter tubing flattened on the sensing end), fast response, total pressure (Ashby, 1988) and static pressure (cone-cylinder) probes. These probes have relatively fast response times because the high-frequency response piezoresistive pressure transducer is located quite close to the sensing orifice. These probes also provide accurate, stable measurements because the transducer is maintained in a controlled thermal environment; that is, housed in a water-cooled jacket. This smaller probe has essentially eliminated probe interference effects observed with larger (0.060) probes. A number of studies have been performed in the 20-Inch Mach 6 Tunnel for which detailed flow field surveys of total pressure were measured. (Static pressures were also measured for several studies.) These studies included wake surveys for very blunt configurations and surveys for slender NASP concepts including detailed pitot pressure (and limited total temperature) surveys in the nozzle flow field with and without blowing of a simulant gas (Everhart, 1992). More recently, steady state pressure and fluctuating temperature measurements in the wake of a spherically blunted cone have been obtained (fig. 12, taken from Horvath et al., 1996). All but one of the AFC tunnels that received upgrades via the F.Y. 1989 major CoF project were equipped with new survey mechanisms.

Data Acquisition/Recording Systems

All hypersonic wind tunnels within the AFC have stand-alone data acquisition systems (DAS). The heart of the system for most tunnels is a 128 or 256-channel, 16 bit, 50 kHz or 100 kHz throughput rate, amplifier per channel, analog-to-digital (A/D) system manufactured by the NEFF Instrument Corporation and having programmable gains and filters per channel and internal clock (system 620/series 600). A typical sampling rate is 50 samples/sec for each channel, although much faster rates are possible. Computers are Hewlett Packard 9000 series, system 745i. Peripheral equipment for each DAS includes dual-disc and optical disc mass storage units, a recordable compact disk unit, and printers, which for several facilities are relatively fast speed (15 lines/sec). All facilities of the AFC have the same hardware and same software to insure compatibility. (As a result of having five compatible systems refinements/enhancements in hardware and software may be developed at one site and downloaded to other sites for incorporation.) These DAS are capable of providing reduced data only minutes after a test.

A separate DAS using Labview-based programs is utilized to display real-time data. This is achieved by monitoring NEFF buffered signal outputs during the run.

Facility-to-Facility Compatibility

As mentioned previously, the five facilities within the AFC are located in three different buildings. As databases are developed, models and portable instrumentation such as strain gage balances and thermographic phosphor systems are routinely transferred from facility to facility. A primary goal in upgrading and enhancing all facility instrumentation with multiple facility testing in mind, is to provide facility to facility compatibility when transferring models. Within each test chamber, instrumentation is situated such that model installation can be accomplished with ease and efficiency. Data acquisition hardware and the data acquisition and reduction software and format are the same for all facilities. As a result test engineers and data acquisition technicians learn the same systems for data acquisition and reduction for all facilities. Output data files follow the same format per specified test requirements and are transferred within minutes following a run to engineering workstations (Sun or MacIntosh) for data analysis. This test set-up philosophy proved to be an invaluable asset in fast-paced assessments of such programs as X-33 Phase 1, X-34 Phase A, and X-38. For these programs aerodynamic and aeroheating databases examining effects of similarity parameters (e.g., viscosity, compressibility, normal shock density ratio or gamma, and wall temperature ratio) were developed in an expeditious manner in order to meet progammatic goals.

Upgrades and Enhancements

Over the past decade, numerous facility upgrades and enhancements have been accomplished via various funding sources, in particular the minor and major CofF programs. The primary purpose for these upgrades and enhancements was to provide improved flow quality and increased capability, reliability, and productivity.

A brief review of CofF, R&D, and maintenance funded projects accomplished to date is illustrated in Table II. Three facilities received new axisymmetric contoured nozzles in the early 1990's. Most of these hypersonic facilities had axisymmetric contoured nozzles designed and fabricated in the late 1950's to early 1970's which produced varying degrees of flow nonuniformity (radially and axially). The 31 Inch Mach 10 Air Tunnel, which has a three-dimensional (i.e., square cross section; all four walls contoured) nozzle, received a new nozzle throat section in order to improve reliability. Figure 13 is an example of the improvement achieved as a result of nozzle redesign and fabrication. (Nozzle design methodology and calibration results for all AFC facilities is presented in Micol, 1995.) Pitot-pressure surveys were measured in all facilities with a rake having probe spacing of only 0.125 in. (Figure 14). Calibration results are presented in Figures 4, 5, 7, 8, 9, and 10.

To protect delicate glass/ceramic thin-film and ceramic phosphor models and to reduce flow turbulence due to the presence of solid particles (contaminants) carried by the freestream flow, 5 to 20 micron in-line filters were installed in all facilities. Facility capabilities were increased with the addition of upgrades such as Flow Field Survey Probes (FFSP), injectable/retractable pitot probes, and improvements to test chambers and model injection and control systems. A new Stokes blower was added to augment eight existing Beech-Russ vacuum pumps for the 15-Inch Mach 6 High Temperature Air Tunnel and 31-Inch Mach 10 Air Tunnel. For the 22-Inch Mach 15/20 Helium Tunnel reclamation system, unreliable pumps and blowers were replaced with two separate trains each consisting of a new Beech-Russ vacuum pump and 3 stage Roots-Connerville blower system.

Facility enhancements initiated in 1997 are presented in Table III, several of which have been completed. An increased capability, augmenting the vacuum system which services both the 31-Inch Mach 10 Air and 15-Inch Mach 6 High Temperature Air Tunnels, was achieved with the addition of a new three stage steam ejector (Fig. 15). This ejector was installed and shakedown initiated in the Spring of 1997. Specifications for the ejector system required evacuation of the two 41-foot diameter spheres and the 60 foot sphere (combined volume equal to 190,000 cu. ft.) from 760 torr to 1 torr in 110 minutes and from 200 torr to 10 torr in 60 minutes. This steam ejector has reduced turn-around time from about 1.5 hours utilizing eight Beech-Russ vacuum pumps to less than 0.75 hour with only the steam ejector operating on the two 41-foot spheres (i.e., a significant increase in pumping capability has been achieved). Operation of both systems simultaneously reduces this recovery time even further. Since the 31-Inch Mach 10 Air Tunnel requires a pre-heat of the electrical resistance heater (i.e., low mass flow through heater) to a temperature of approximately 2000 °R to eliminate thermal shock to the heater tube bundle, the addition of the steam ejector has now made possible simultaneous pre-heat and sphere evacuation continues. (Preheat of the 15-Inch Mach 6 High Temperature Air Tunnel is performed to atmosphere; therefore, simultaneous pre-heat and evacuation is always accomplished). As a result of this enhanced capability, productivity has increased by approximately 1.5 to 2.0 times for both facilities and, because of the lower vacuum and higher evacuation capacities, run times for each facility have increased also.

As discussed in Micol, 1995, the 15-Inch Mach 6 High Temperature Air Tunnel is a conversion of a facility originally designed to perform aerodynamic flutter testing at Mach 10 in air. As a Mach 10 facility, a 1.25 MW electrical heater was employed to heat air to 1500 °R; however, this temperature level was insufficient to avoid liquefaction during expansion to Mach 10 for reservoir pressures matching those of the 31-Inch Mach 10 Air Tunnel. Primarily because of the need to match unit free-stream Reynolds number, R_∞, ratio of specific heats, γ_∞, and ratio of wall-to-adiabatic wall temperatures, T_w/T_{aw}, in order to determine compressibility or Mach number effects for slender configurations, the facility was modified. The existing Mach 10 nozzle was modified to provide Mach 6 flow at the same R_∞, γ_∞ and nearly the same T_w/T_{aw} as the 31-Inch Mach 10 Tunnel.

A number of enhancements to the 15-Inch Mach 6 High Temperature Air Tunnel were performed in the Spring of 1995. The 1.25 MW heater was upgraded to 5 MW in order to handle higher mass flow rates (i.e., 20 lbm/sec at the highest reservoir pressure setting), a walk-in test chamber identical to that of the 20 Inch Mach 6 CF_4 was installed, and a 5 micron in-line filter was added.

As expected, the original diffuser performed poorly at the new Mach 6 condition. The original second minimum utilized a variable area section, whereby area change was achieved by axially translating a conical plug (Hodge, 1992). As shown in Fig.16, the original diffuser which was designed for Mach 10 operation possessed a catch cone with a 8° half angle, a variable diameter duct having a minimum diameter of 13 inches followed by a conical section which increased in diameter to 15 inches, in which the conical plug transversed. The plug was positioned to a fixed location during calibrations of the Mach 10 nozzle and during the conversion to Mach 6, to provide the maximum possible flow-through area. (At this time, the facility had a closed jet test section.) Conversion of the closed jet test section to a significantly larger open jet test section revealed a problem with the diffuser's ability to process the fluid, thus producing partial and often total blockage for most test articles. The conical plug was removed and replaced with a smaller conical section of significantly less blockage area. During nozzle calibration with the open jet test section, an unknown event or change in conditions prevented the tunnel from remaining started with the pitot rake injected. No cause could be found for the anomaly, but diffuser performance was suspected. The diffuser was dismantled, modified, and reassembled with a larger flow-through area fixed diffuser. Following this conversion, the tunnel was operated with marginal success for a limited set of configurations due to the occurrence of tunnel blockage.

A FY 1996 CofF was proposed to provide a new variable area diffuser and heat exchanger for several reasons: 1) reduce blockage effects and improve pressure recovery by properly sizing the new diffuser for optimal performance at Mach 6, 2) improve diffuser performance for larger models by requiring the diffuser to be variable area (i.e., a translating plug as opposed to a moveable wall), and 3) replace the antiquated and unreliable tube sheet type heat exchanger with newer technology (Fig.17).

The initial design approach for a new diffuser was to model the existing diffuser (i.e., the Mach 10 diffuser) using computational fluid dynamic (CFD) codes and compare predicted to measured static pressures along the diffuser wall. Once the code was validated for the existing diffuser, aerolines for a new supersonic diffuser with a fixed second minimum would be calculated. As a result of time constraints and the inability to achieve a converged CFD solution, this approach was abandoned.

The approach used to design the diffuser was based on published experimental results and "rules of thumb" guidelines of other NASA and AEDC tunnels (see for example, Midden et al., 1964; Andrews, 1994; Austin, 1966). Also, a 17 inch diameter constant area duct was fabricated, instrumented and tested as a means of examining the effect of a significant increase in flow through area on tunnel performance. Static pressure ports were located at 6 inch intervals along it's length and total pressure was measured at three locations. Comparisons of measured pressure data from this "research" diffuser to published results (Austin, 1966) were performed. Testing of this research diffuser revealed an improvement in the operational performance of the tunnel; however, tunnel run time was decreased and an inability to operate at the nozzle design pressure and temperature (i.e., $P_{t,1}$=45 psia; $T_{t,1}$= 1260 °R) occurred as pressure recovery decreased.

Confidence, bolstered by the research diffuser's performance, and good agreement between published and measured data led to the development of the design shown in Fig. 17. Based on these published results and "rules of thumb" a minimum starting diameter of 16 inches was proposed. Flexibility was incorporated into the design to allow an increase to a maximum diameter of 17 inches if required. As shown in Fig. 17, a translating conical centerbody plug is used to vary the area. This centerbody can be moved approximately 32 inches in 10 seconds, thus decreasing the second minimum area from 193 in.^{2} at full retracted position to 102.5 in.^{2} at the most forward position. Fig.18 illustrates the effect of reducing the second minimum area on run time. For the tunnel empty (no model in test section), operational time is increased by a factor of 1.25 when the probe is translated forward to full stroke. Tunnel operation with models of varying size (3 to 5 inch diameter) and shapes (cones, spheres, lifting bodies, etc.) indicate that a centerbody location of 12 to 15 inches is satisfactory to prevent blockage and provide sufficient run time for force and moment and pressure tests. Blockage tests are required for each test article to define the optimum operational envelope for both model and tunnel (i.e., range of angle of attack/model parametrics to prevent blockage and optimum centerbody location for model size and geometry).

To improve optical access for video based non-intrusive diagnostic methods in the 31-Inch Mach 10 Air Tunnel, a 1997 CofF project was initiated. This CofF project deemed "31-Inch Mach 10 Optical Access" provides for optical access of the test section from three locations (see fig. 19), thus permitting simultaneous viewing for optical video-based techniques. This enhancement will reduce the number of runs required since assessment of a configuration's aeroheating and aeroloads characteristics from several viewports for a given angle of attack may be achieved in a single run. The aquisition of two/three-view data increases both efficiency and productivity. Also, included in this CofF is the relocation of the facility's Flow Field Survey Probe from a location beneath the test section to the top, a new injectable-retractable pitot-probe, replacement 5 micron in-line filter, and a model preparation area.

Design has been completed for a FY 1998 CofF project for the 20-Inch Mach 6 Air Tunnel. This project includes fabrication of a new stainless steel settling chamber and the addition of rigimesh screens and a stainless steel perforated cone with rigimesh covering, to reduce acoustic noise, (see Dillon et al., 1988, Beckwith, 1981, and Beckwith et al., 1986, 1988, 1990); and stainless steel screen holders housing inconel screening. A follow-on activity currently in design is the replacement of the Mach 6 two-dimensional nozzle, potted and gouged from years of use, with a new two-dimensional nozzle fabricated from Invar, a low coefficient of thermal expansion material. This new nozzle extends into the test section where improved optical accesses will be incorporated. Table IV identifies the particular area of improvement for current/future upgrades.

Future Facility Upgrades and Enhancements

A study was conducted to identify a replacement heater system for the 20-Inch

Mach 6 CF_4 Tunnel. Currently, a storage-bath heater system is employed. A new system is desired to enhance temperature control, decrease the transient to required temperature set points, and improved productivity and reliability. An analysis has shown that a natural gas heater system is capable of meeting these requirements.

Modifications to the reclamation systems for both the 22-Inch Mach 15/20 Helium and 20-Inch Mach 6 CF4 are proposed which will improve productivity, and reliability.

Instrumentation Enhancements

In the area of instrumentation, the future emphasis will continue to be on the advancement of optical, video-based thermography systems. Thermography systems now provide essentially all of the aeroheating information from the AFC, thus complimenting aerodynamic performance with aeroheating characteristics in a timely fashion. These systems, taking advantage of advances in data imaging and acquisition, in computer and mass storage technology, and in software development will provide increasingly more accurate quantitative, global two-dimensional (existing capability) and three-dimensional heating distributions via the simultaneous utilization of multiple cameras or a mirror system. (The feasibility of simultaneously obtaining thermal mappings on the windward and leeward surfaces of a model at Mach 10 has been demonstrated by Daryabeigi.)

Optical techniques providing global pressure distributions (e.g., DeMeis, 1992, Buck, 1994), similar to thermal mapping techniques, will be transitioned from the laboratory to the wind tunnel and become widely and routinely used. Continuous development of luminescent diagnostic systems for simultaneous pressure and temperature surface measurements has occurred since 1994 (see Buck, 1994). Evolvement of this system includes the development of a new higher temperature coating (to 275 Celsius or 530° Fahrenheit), a new means of phosphor excitation, and an improved imaging system. A new high-temperature luminescent material (PT600) and coating system has been developed which provides durable, uniform, and smooth coatings on ceramic and metal surfaces. Using an advanced imaging and pulsed laser excitation system, luminescent coatings with PT600 provide single-pulse measurement of pressure and temperature simultaneously. Demonstration of this new technology has been performed in laboratory experiments. Current development is focused on refining methods for model coating and measurements in hypersonic blow-down wind tunnels. Completion and demonstration of this technology is expected by mid-1998. (Buck, 1998)

Ceramic models will, to a large degree, replace metal models for most force and moment and pressure studies, as they already have for heat transfer studies.

Recently quantitative planar measurements of density were obtained in the 15-Inch Mach 6 High Temperature Air Tunnel utilizing the Rayleigh scattering technique (Shirinzadeh et al., 1996). Signals generated using a narrow-band pulse ArF excimer laser were detected using a single-intensified CCD camera. Measurements were performed in the freestream, behind the bow shock, and in the base region and wake of a cylinder. The high stagnation temperature capability of the 15-Inch HTA (1400 °R) makes quantitative measurements possible over a reservoir pressure range from 50 to 500 psia, thus eliminating the clustering effects observed by Shirinzadeh et al., 1991 at lower temperatures. Good agreement between CFD density calculations and measurement was achieved (Shirinzadeh et al., 1996). With further enhancemnents to the experimental model and tailoring of facility operating conditions, it may be possible to obtain these same measurements with reduced uncertainty and, perhaps, on a routine basis.

Facility Utilization

As a result of several high priority, fast-paced Agency programs to design and fly experimental vehicles in the 1998/1999 timeframe, and the corresponding requirements for aerodynamic and aeroheating information, the AFC facilities have been heavily utilized since the spring of 1995. These programs include:

1) X-33 Phase I - Reuseable Launch Vehicle/Advanced Technology Demonstrator - Three different Industry concepts assessed and optimized simultaneously

2) X-33 Phase II - Optimization/ benchmarking for the Lockheed Martin Skunkworks configuration

3) X-34 Phase B - Small Reuseable Technology Demonstrator - under the auspices of RLV program

4) Hyper X - A Mach 5, 7, and 10 airbreathing propulsion flight experiment

5) Experimental Crew Return Vehicle, X-CRV/X-38

6) Orbiter Enhancements

7) Planetary - Mars Microprobe, Mars Sample Return, etc.

The heaviest utilization of the AFC occurred during the RLV/X-33 Phase I, X-34, and X-38 programs, whereby aerodynamic and aeroheating information was generated to assess and optimize the performance of five industry concepts in parallel. For the initial aerodynamic screening of X-33 concepts, the 22-Inch Mach 15/20 Helium Tunnel was a major contributor. Because of the ability to manufacture plastic stereo-lithography models (fig. 20) in a matter of days, configuration geometries could be rapidly assessed and modified if necessary because of this tunnel's ability to operate at ambient conditions (i.e., with plastic or even wooden models). The outer mold lines of the X-33 configurations were often altered in "real time" to examine parametric changes. Approximately 400 runs were performed for the RLV/X-33 concepts in this facility for Phase I, 140 runs for Phase II, and it is presently being used to assess the aerodynamic performance of LMSW RLV concepts.

The use of the thermographic phosphor technique to establish aeroheating databases has revolutionized the process to generate large aerothermal databases rapidly. For example, in X-33 Phase I, for each aerodynamic database developed, a complementary aeroheating database was also acquired (see figs. 6 and 21), providing the configuration designers with aerodynamic performance and aeroheating information for assessment and optimization of their concepts.

Concluding Remarks

The Langley Aerothermodynamics Facilities Complex consists of five hypersonic, blowdown-to-vacuum wind tunnels developed and initially operated between the late 1950's and mid-1960's. These tunnels provide essentially all of the Agency's aerothermodynamic testing capability. They compliment one another to provide ranges of Mach number from 6 to 20, unit Reynolds number from 0.05 to 25 million per ft, and normal shock density ratio from 4 to 12 (or, specific heat ratio within the shock layer of models from 1.10 to 1.67). Several facilities provide unique capabilities, primarily because of the test gas.

The present paper basically represents a compilation of the following information:

(1) Description of the basic components of each facility, capability in terms of reservoir pressure and temperature and corresponding range of free stream flow conditions.

(2) Instrumentation and testing techniques routinely used to measure forces and moments, surface pressures, surface temperature-time histories (heat transfer rates), flow properties within the shock layer about the model via survey probes (i.e., intrusive techniques), and flow visualization (schlieren, oil flow patterns, etc.); also included are descriptions of data acquisition systems.

(3) Description of AFC facility upgrades performed since 1995 with a discussion of future approved and proposed facility enhancements.

(4) Recent utilization of AFC facilities for several high priority fast-paced Agency programs (e.g., X-33 and X-34).

These facilities provide an excellent capability for parametric aerodynamic and aeroheating studies required in the early design and assessment of proposed advanced aerospace vehicles and currently are being used to provide benchmarking data as well.

Acknowledgments

The author wishes to acknowledge the contributions of Messrs. G. M. Buck, R. N. Merski; M. R. Rhode; C. E. Glass; and C. G. Miller; Head, Langley Research Center's Aerothermodynamics Branch, Aero- and Gas-Dynamics Division, R. J. Balla, Measurement Science and Technology Branch, Fluid Mechanics and Acoustics Division, and J. W. Allred, Facilities Engineering Branch, Facility Systems Engineering Division, for their technical advice in the preparation of this paper.

References

Andrews, E. H., "Nozzle/Tunnel Starting in Free-Jet Engine Test Facilities: A Bibliography." CPIA Publication 585, 1994.

Ashby, G. C., Jr., "Miniaturized Compact Water-Cooled Pitot-Pressure Probe for Flow-Field Surveys in Hypersonic Wind Tunnels." ISA 34th International Instrumentation Symposium, 1988.

Austin, R. F., "An Open-Jet Wind Tunnel Investigation of a Fixed-Geometry Diffuser System at Mach Numbers 3.6 and 7.0." AEDC-TR-66-11, 1966.

Beckwith, I. E., "Comments on Settling Chamber Design for Quiet, Blowdown Wind Tunnels." NASA TM 81948, March 1981.

Beckwith, I. E., Chen, F. J., and Malik, M. R., "Design and Fabrication Requirements for Low-Noise Supersonic/Hypersonic Wind Tunnels." AIAA Paper No. 86-0763-CP, 1986.

Beckwith, I. E. Chen, F. J., and Malik, M. R., "Design and Fabrication Requirements for Low-Noise Supersonic/Hypersonic Wind Tunnels." AIAA Paper No. 88-0143.

Beckwith, I. E., Chen, F. J., Wilkinson, S. P., Malik, M. R., and Tuttle, D. G., "Design and Operational Features of Low-Disturbance Wind Tunnels at NASA Langley for Mach Numbers from 3.5 to 18." AIAA Paper 90-1391.

Benton, James R., "Design and Navier-Stokes Analysis of Hypersonic Wind Tunnel Nozzles," M.S. thesis, North Carolina State University, Department of Mechanical and Aerospace Engineering, 1989.

Benton, J. R., Perkins, J. N., and Edwards, A. C., "Limitations of the Method of Characteristics When Applied to Axisymmetric Hypersonic Nozzle Design." AIAA Paper 90-0192, January1990.

Berry, S. A., et al., "Results of the 0.0362-Scale X-38 (Rev 3.1) Vehicle Aerothermodynamic and Boundary Layer Transition Test in the NASA Langley 20-Inch Mach 6 Tunnel." NASA TM 112857, 1997.

Berry, S. A., et al., "X-34 Experimental Aeroheating at Mach 6 and 10." AIAA 98-0881.

Buck, G. M., "An Imaging System for Quantitative Surface Temperature Mapping Using Two-Color Thermographic Phosphors." ISA 34th International Instrumentation Symposium, 1988.

Buck, G. M., "Surface Temperature/Heat Transfer Measurement Using a Quantitative Phosphor Thermography System." AIAA 91-0064, 1991.

Buck, G. M., "Simultaneous Luminescence Pressure and Temperature Measurements on Dyed Ceramic Models for Hypersonic Wind Tunnels," AIAA Paper 94-2482, June 1994.

Buck, G. M., personal communication, 1998.

Burner, A. W., and Midden, R. E., "Holographic Flow Visualization at the Langley CF_4 Tunnel," NASA TM-74051, July1977.

Braun, R. D., et al., "Mars Pathfinder Six Degree-of-Freedom Entry Analysis," Journal of Spacecraft and Rockets, Vol. 32, No. 6, Nov.-Dec. 1995, pp. 993-1000.

Campbell, C., et al., "Overview of X-38 Hypersonic Wind Tunnel Data and Comparison with Numerical Results." AIAA Paper 97-0567.

Campbell, C., et al., "An Overview of X-38 Hypersonic Aerothermodynamic Wind Tunnel Data and Comparison with Numerical Results." AIAA Paper 97-2475.

Colladay, R. S., "Rekindled Vision of Hypersonic Travel." Aerospace America, pp. 30, 31, and 34, August1987.

Compton, E. C., and Spencer, R. C., "Guide-lines for Schlieren Systems at LRC," LWP-448, July1967.

Cook, W. J., and Felderman, E. J., "Reduction of Data from Thin-Film Heat Transfer Gages: A Concise Numerical Technique." AIAA Journal, Vol. 4, pp. 561-562, March1966.

Daryabeigi, K., "Global Surface Temperature/ Heat Transfer Measurements Using Infrared Imaging." AIAA 92-3959, 1992.

DeMeis, R., "Paint Under Pressure." Aerospace America, pp.40-41, March1992.

Dillon, J. L., Trimpi, R. L., and Wilcox, F. J. Jr. "Unexpected/Expected Results frokm the Langley 20-Inch Supersonic Wind Tunnel During Initial Chekout." AIAA 88-1999 CP.

Everhart, J. L., Ashby, G. C., Monta, W. J., "Preliminary Nozzle Pitot-Pressure Surveys on Langley Test Technique Demonstrator at Mach 6." AIAA 92-3940, 1992.

Gnoffo, P. A., et al., Influence of Sonic Line Location on Mars Pathfinder Aerodynamics." Journal of Spacecraft and Rockets, Vol. 33, No. 2, Mar.-Apr. 1996, pp. 169-177.

Gnoffo, P. A., et al., "Computational Aerothermodynamic Design Issues for Hypersonic Vehicles." AIAA Paper 97-2473.

Gnoffo, P. A., et al., "Prediction and Validation of Mars Pthfinder Hypersonic Aerodynamic Data Base," to be presented at the 7th AIAA/ASME Joint Thermophysics and Heat-Transfer Conference, June 1998.

Goldberg, T. J., and Hefner, J. N., "Starting Phenomena for Hypersonic Inlets with Thick Turbulent Boundary Layers at Mach 6." NASA TN D-6280, August1971.

Hamilton, H. H. III, et al., "Computational/Experimental Aeroheating Predictions for X-33 Phase II Vehicle." AIAA Paper 98-0869.

Harvey, W. D., and Hunter, W. W., "Experimental Study of a Free Turbulent Shear Flow at Mach 19 with Electron-Beam and Conventional Probes." NASA TN D-7981, October1975.

Hodge, J. S., "Langley 15-Inch Mach 6 High Temperature Tunnel." AIAA 92-3938, 1992.

Hollis, B. R.., "User's Manual for the One-Dimensional Hypersonic Experimental Aero-Thermodynamic (1DHEAT) Data Reduction Code." NASA CR, 1995.

Hollis, B. R.., "Real-Gas Flow Properties for NASA Langley Research Center Aerothermodynamic Facilities Complex Wind Tunnels." NASA CR, 4755.

Honaker, W. C., et al., "A Study of Density Measurements in Hypersonic Helium Tunnels Using an Electron Beam Fluorescence Technique." AIAA Paper 79-1085, June 1979.

Horvath, T. J. et al., "Blunt Body Near Wake Flow Field at Mach 6." AIAA Paper 96-1935.

Horvath, T. J. et al., "Blunt Body Near Wake Flow Field at Mach 10." AIAA Paper 97-0986.

Jones, R. A., and Hunt, J. L., "Use of Tetrafluoromethane to Simulate Real-Gas

Effects on the Hypersonic Aerodynamics of Blunt Vehicles." NASA TR R-312, June1969.

Jones, J. J., "The Rationale for an Aeroassist Flight Experiment." AIAA Paper No. 87-1508, 1987.

Kendall, D. N., "Semiconductor Surface Thermocouples for Determining Heat-Transfer.", IEEE Tranactions on Aerospace and Electronic Systems, Vol. AES-3, No. 4, July, 1967, pp. 596 - 603.

Loomis, M. P., et al., "Aeroheating and Aerodynamic CFD Validation and Prediction for the X-38 Program." AIAA Paper 97-2475.

Merski, N. R., "A Relative - Intensity Two-Color Phosphor Thermography System." NASA TM-104123, 1991.

Merski, N. R., "Reduction and Analysis of Phosphor Thermography Data with the IHEAT Software Package." AIAA 98-0712.

Micol, J. R., Midden, R. E., and Miller, C. G., "Langley 20-Inch Hypersonic CF_4 Tunnel: A Facility for Simulating Real Gas Effects." AIAA 92-3939, 1992.

Micol, J. R., "Simulation of Real-Gas Effects on Pressure Distributions for Aeroassist Flight Experiment Vehicle and Comparison with Prediction." NASA TP-3157, 1992.

Micol, J. R., "Hypersonic Aerodynamic/Aerothermodynamic Testing Capabilities at Langley Research Center: Aerothermodynamics Facilities Complex." AIAA Paper No. 95-2107.

Midden, R. E., and Cocke, B. W. Jr., "Diffuser Performance of a Mach 6 Open-Jet Tunnel and Model-Blockage Effects at Stagnation Temperatures to 3,600 °F." NASA TN D-2384, July1964.

Midden, R. E., and Miller, C. G., "Description and Calibration of the Langley Hypersonic CF_4 Tunnel." NASA TP-2384, March1985.

Miller, C. G., and Gnoffo, P. A., "Pressure Distribution and Shock Shapes for 12.84°/7° On-Axis and Bent-Nose Biconics in Air at Mach 6." NASA TM-83222, December1981.

Miller, C. G., "Experimental and Predicted Heating Distributions for Biconics at Incidence in Air at Mach 10." NASA TP-2334, November1984.

Miller, C. G., et al., "Laminar Heat-Transfer Distributions on Biconics at Incidence in Hypersonic-Hypervelocity Flow." NASA TP-2213, January1985.

Miller, C. G., "Langley Hypersonic Aerodynamic/ Aerothermodynamic Testing Capabilities-Present and Future." AIAA Paper No. 90-1376, 1990.

Miller, C. G., "Hypersonic Aerodynamic/Aerothermodynamic Testing Capabilities at Langley Research Center." AIAA Paper No. 92-3937, 1992.

Mitcheltree, R. A., et al., "Aerothermal Heating Predictions for Mars Microprobe." AIAA Paper 98-0170.

Naftel, J. C., Powell, R. W., Talay, T. A., "Ascent, Abort, and Entry Capability Assessment of a Space Station Rescue and Personnel/ Logistics Vehicle." AIAA Paper No. 89-0635, 1989.

Neumann, R. D., "Aerothermodynamic Instrumentation." AGARD Report No. 761, pp. 4-1 to 4-40, 1988.

Paulson J.W., Jr. and Braukmann, G.J., "Recent Ground-Facility Simulations of Space Shuttle Orbiter Aerodynamics." Orbiter Experiments (OEX) Aerothermodynamics Symposium, Williamsburg, Virginia, April 27-30, 1993, published in Orbiter Experiments (OEX) Aerothermodynamics Symposium, NASA Conference Publication 3248, Part 1, April 1995.

Piland, W. M., and Talay, T. A., "Advanced Manned Launch System Comparisons." IAF-89-221, October1989.

Schultz, D. L., and Jones, T. V., "Heat-Transfer Measurements in Short-Duration Hypersonic Facilities." AGARD AG-163, February1973.

Shirinzadeh, B., et al., "Study of Cluster Formation and Its Effects on Rayleigh and Raman Scattering Measurements in a Mach 6 Wind Tunnel." AIAA 91-1496, 1991.

Shirinzadeh, B., et al., "Rayleigh Scattering Measurements Supersonic Facilities." AIAA 96-2187.

Wagner, R. D., and Weinstein, L. M., "Hot-Wire Anemometry in Hypersonic Helium Flow." NASA TN D-7465, June1974.

Walberg, G. D., "A Review of Aeroassisted Orbit Transfer." AIAA Paper 82-1378, August1982.

Walberg, G. D., "A Review of Aerobraking for Mars Missions." IAF 88-196, 1988.

Woods, W. C., and Arrington, J. P., "Electron-Beam Flow Visualization - Applications in the Definition of Configuration Aerothermal Characteristics." AIAA Paper 72-1016, September1972.

Table 1. Representative flow conditions.

Facility	Test Gas	$p_{t,1}$, psi	$T_{t,1}$, °R	p_∞, psi x10^2	T_∞, °R	q_∞, psi	V_∞, ft/s	M_∞	R_∞, ft^{-1} x10^{-6}	R_2, ft^{-1} x10^{-5}	ρ_2/ρ_∞ psi	$p_{t,2}$, psi	Test Core Size, in.
15-Inch Mach 6 Hi Temp	Air	51	930	3.166	112.6	0.804	3139	6.03	0.784	1.340	5.29	1.494	10
		104	922	6.323	111.2	1.619	3127	6.05	1.610	2.725	5.30	3.008	10
		203	918	12.285	110.6	3.151	3119	6.06	3.161	5.336	5.30	5.855	10
		347	900	20.559	108.0	5.311	3087	6.09	5.529	9.214	5.30	9.867	10
		415	961	24.461	115.0	6.336	3194	6.09	5.963	10.159	5.31	11.774	10
		53	1224	3.1901	149.3	0.816	3622	6.04	0.518	0.981	5.36	1.519	10
		104	1253	6.4153	154.1	1.629	3666	6.02	0.990	1.903	5.36	3.031	10
		197	1266	12.142	155.8	3.086	3686	6.02	1.845	3.561	5.36	5.742	10
		362	1261	21.528	153.4	5.544	3682	6.06	3.370	6.420	5.37	10.314	10
		445	1268	26.344	154.0	6.800	3693	6.07	4.102	7.819	5.37	12.648	10
20-Inch Mach 6	Air	29	868	2.140	111.6	0.514	3020	5.86	0.533	0.934	5.25	0.956	13x12
		59	882	4.238	111.7	1.028	3047	5.88	1.046	1.830	5.26	1.910	13x12
		124	922	8.120	113.7	2.030	3122	5.98	1.980	3.422	5.28	3.772	13x13
		250	911	16.116	111.8	4.051	3103	5.99	4.043	6.933	5.28	7.528	14x14
		366	936	23.111	114.2	5.852	3142	6.02	5.632	9.724	5.29	10.876	14x14
		476	930	29.813	113.1	7.573	3136	6.04	7.383	12.651	5.29	14.073	14x16
31-Inch Mach 10	Air	348	1800	0.992	95.2	0.650	4628	9.67	0.568	0.472	5.96	1.205	10x10
		723	1825	1.867	94.2	1.259	4670	9.81	1.104	0.898	5.98	2.334	12x12
		1452	1800	3.509	90.7	2.430	4643	9.94	2.240	1.755	5.98	4.503	12x12
20-Inch Mach 6	CF_4	87	1124	0.400	319.6	0.086	2782	5.90	0.040	0.136	11.46	0.168	12
		107	1129	0.467	319.2	0.101	2792	5.92	0.047	0.160	11.50	0.198	12
		156	1129	0.633	314.6	0.139	2797	5.97	0.066	0.220	11.54	0.273	12
		512	1177	1.977	337.1	0.428	2872	5.95	0.183	0.640	11.71	0.840	14
		956	1167	3.642	328.6	0.799	2854	5.98	0.353	1.212	11.68	1.568	14
		1523	1131	5.590	303.1	1.276	2796	6.06	0.625	2.025	11.59	2.500	14
		1935	1168	7.287	324.4	1.619	2853	6.01	0.725	2.458	11.70	3.173	14
22-Inch Mach 15/20	He	494	528	1.409	8.0	2.296	5702	13.98	4.777	2.909	3.94	4.053	10
		695	528	1.898	7.9	3.156	5710	14.12	6.665	3.986	3.94	5.571	10
		1185	528	2.928	7.6	5.102	5731	14.46	11.157	6.393	3.94	9.006	12
		1700	528	3.903	7.4	7.054	5752	14.73	15.809	8.767	3.94	12.450	12
22-Inch Mach 15/20	He	300	528	0.270	5.1	0.700	5711	17.64	2.400	0.887	3.96	1.235	5
		502	528	0.349	4.6	1.007	5721	18.61	3.875	1.271	3.97	1.776	7
		1007	528	0.503	4.0	1.670	5744	19.95	7.372	2.090	3.97	2.945	9
		1512	528	0.629	3.7	2.260	5765	20.77	10.685	2.807	3.97	3.987	10
		2429	528	0.854	3.5	3.325	5802	21.60	16.573	4.070	3.97	5.864	11
		3309	528	1.041	3.3	4.284	5837	22.21	22.013	5.172	3.98	7.555	11

Table II. Completed AFC facility upgrades and improvements.

Modification	15" M-6 HT Air	20" M-6 Air	31" M-10 Air	20" M-6 CF_4	22" M-15/ 20 He
Nozzle	X		throat	X	X
Filter	X	X	X	X	X
FFSP		X	X	X	X
Pitot probe		X		X	X
Test section	X			X	X
Model injection	X	X		X	X
Hi pressure	X				
Vacuum					X
Fund source	R&D NASP '91 CoF '93 CoF '96 CoF	'88 CoF '89 CoF	'89 CoF '91 CoF '96 CoF '97 CoF	'88 CoF '89 CoF	'88 CoF '89 CoF '92 CoF '93 CoF R&D/ maintenance

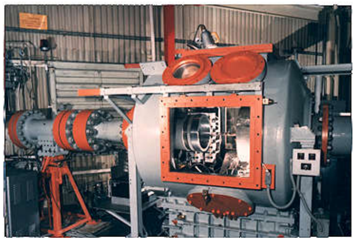
Table III. Recent tunnel hardware upgrades and modifications.

Modification	15" M-6 HT Air	20" M-6 Air	31" M-10 Air	20" M-6 CF_4	22" M-15/20 He
Steam Ejector	X		X		
Variable Area Diffuser	X				
Heat Exchanger	X				
Heater Modification	X			X	
Nozzle		X			
Settling Chamber		X			
Reclaimer Mods				X	X
DAS Upgrades	X	X	X	X	X

Table IV. Enhancement categories for current/future upgrades.

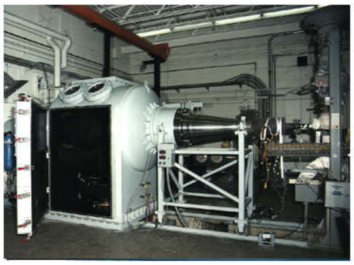
Tunnel	Capability	Productivity	Reliability
15-Inch Mach 6 HTA	Variable area diffuser Heater mods Test section	Steam ejector DAS upgrade	Heat exchanger
20-Inch Mach 6 Air	Settling chamber *Steam ejector *Schlieren system	DAS upgrade	Nozzle
20-inch Mach 6 CF_4	New heater	DAS upgrade	N_2 Dewar Heater mods Reclaimer mods
31-Inch Mach 10 Air	Optical access Pitot probe	Steam ejector DAS upgrade	Heater mods Filter
22-Inch Mach 15/20 He	Electron beam	Add'l vacuum Add'l reclaimer DAS upgrade	New vacuum capability

*Improvements of existing capability

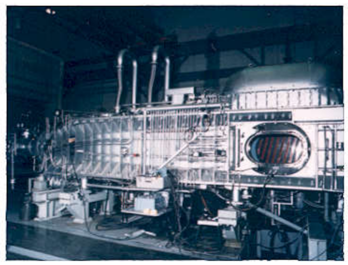


15-Inch Mach 6 Hi Temp. Air

20-Inch Mach 6 Air



20-Inch Mach 6 CF_4



31-Inch Mach 10 Air

22-Inch Mach 15/20 He

Fig. 1 Facilities of the Aerothermodynamic Facilities Complex.

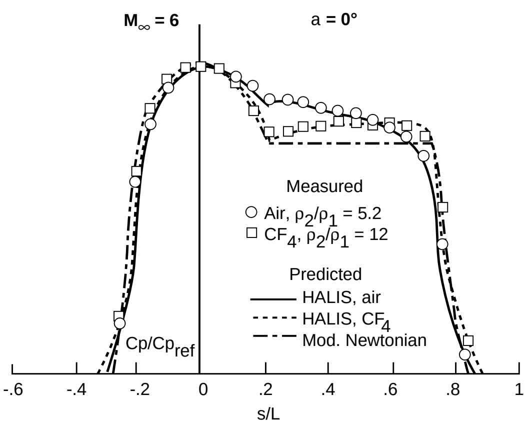


Fig. 2 (a) Effect of normal shock density ratio on AFE symmetry plane pressure distributions at M_∞ = 6 and α = 0° (from Micol, 1992).

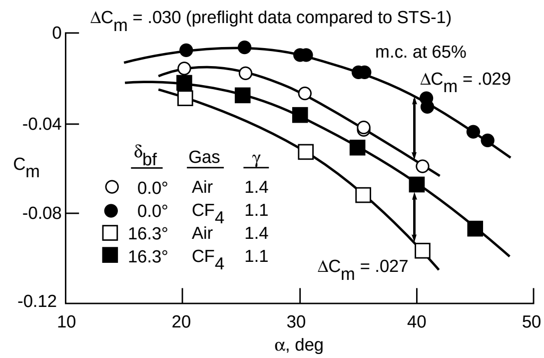


Fig. 3 Effect of gamma on 0.004 scale Space Shuttle Orbiter pitching-moment characteristics measured in 15-Inch Mach 6 Air and 20-Inch Mach 6 CF_4 Tunnels, M_∞ = 6 and δ_{bf} = 0° and 16.3°.

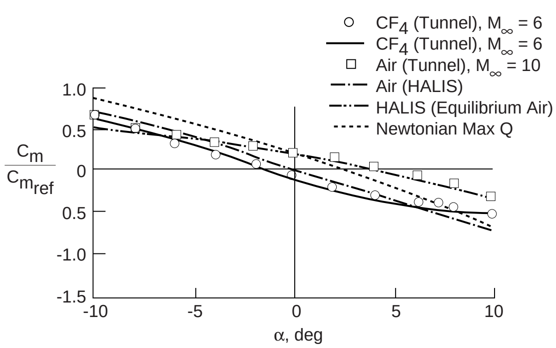


Fig. 2 (b) Measured and predicted pitching moment characteristics for wind tunnel conditions and maximum dynamic pressure flight case (from Micol, 1992).

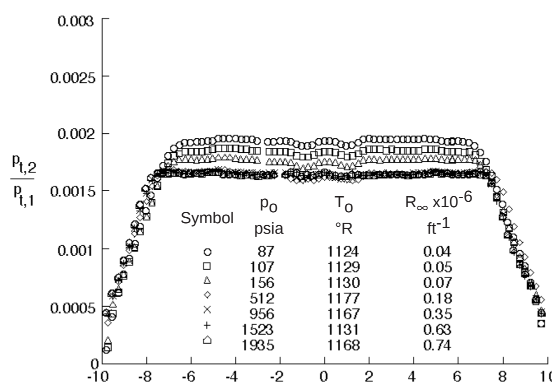


Fig. 4 Pitot-pressure profiles at nozzle exit for various reservoir pressures in 20-Inch Mach 6 CF_4 Tunnel (from Micol 1992).

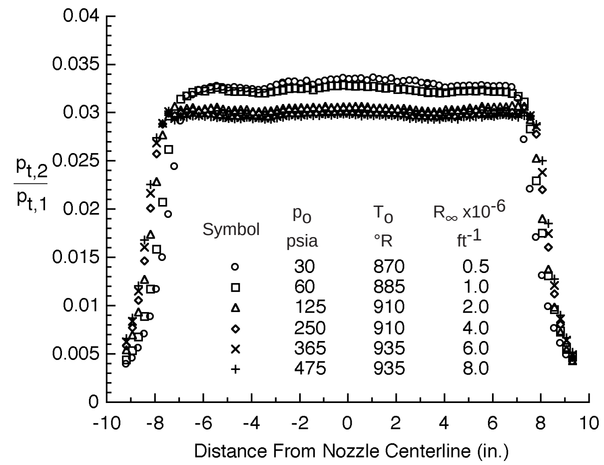


Fig. 5 Pitot pressure profiles at nozzle exit for various reservoir pressures in 20-Inch Mach 6 Air Tunnel (from Micol, 1995).

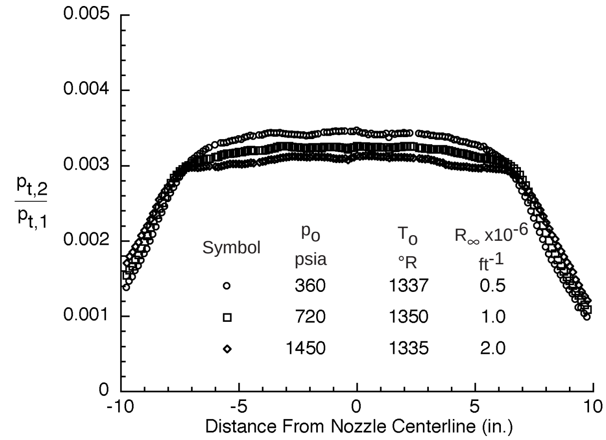


Fig. 8 Pitot-pressure profiles measured in 31-Inch Mach 10 tunnel for various reservoir pressures (from Micol, 1995).

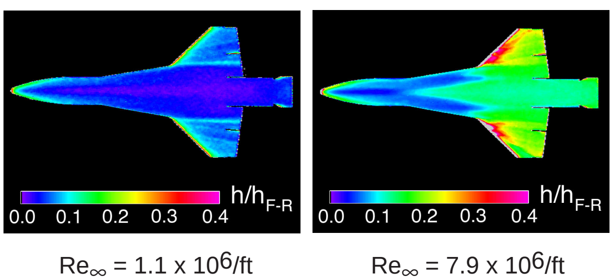


Fig. 6 Effect of Reynolds number on windward heating rates for X-34 at $M_\infty = 6$, $\alpha = 0°$, and $\delta_{CS} = 0°$.

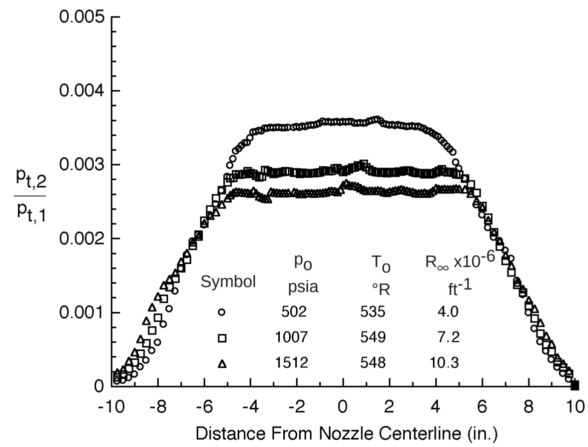


Fig. 9 Pitot-pressure profiles measured in the 22-Inch Mach 15/20 Helium Tunnel with Mach 20 nozzle for various reservoir pressures (from Micol, 1995).

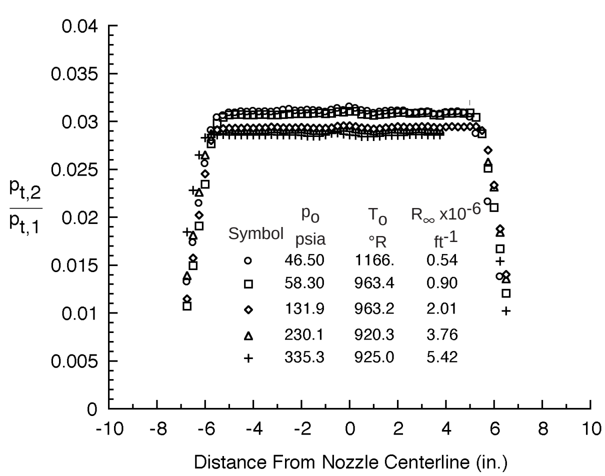


Fig. 7 Pitot-pressure profiles measured in 15-Inch Mach 6 High Temperature Air Tunnel for various reservoir pressures (taken from Hodge, 92).

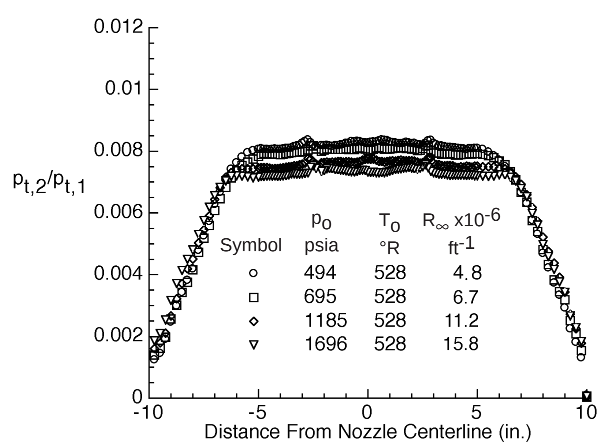


Fig. 10 Pitot-pressure profiles measured in 22-Inch Mach 15/20 Helium Tunnel with Mach 15 nozzle for various reservoir pressures.

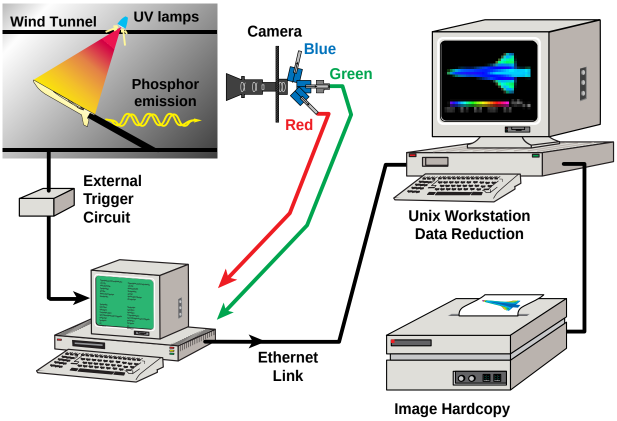


Fig. 11 Illustration of two-color relative intensity phosphor thermography acquisition system.

Fig. 14 Photograph of 157 probe pitot-pressure rake; rake probe spacing equals 0.125 in.

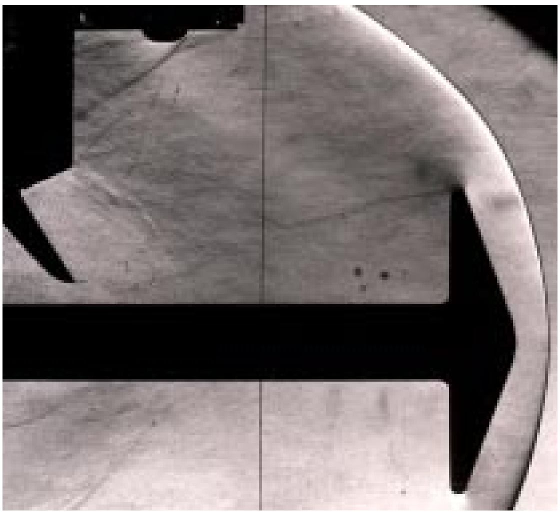


Fig. 12 Wake survey studies of spherically blunted cone model in 20-Inch Mach 6 Air Tunnel.

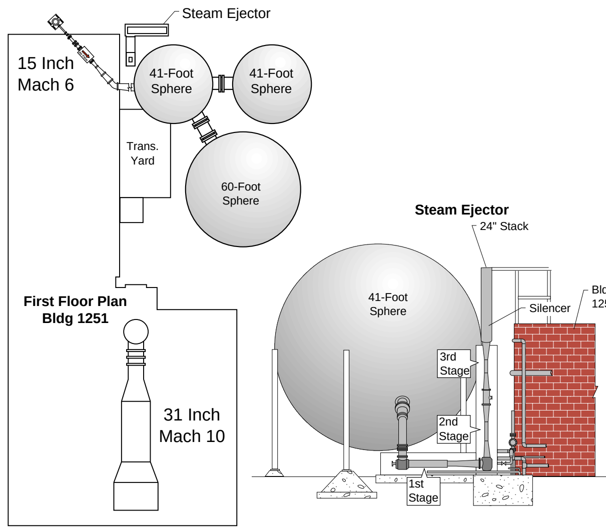


Fig. 15 Sketch of three-stage steam ejector augmenting vacuum capability for 31-Inch Mach 10 and 15-Inch Mach 6 HT Air Tunnels.

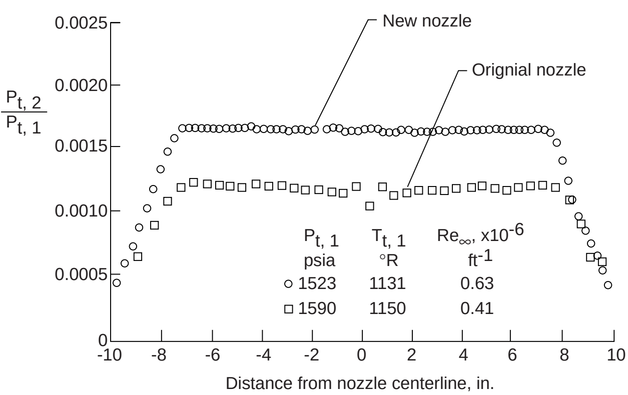


Fig. 13 Comparison of pitot-pressure distributions measured with original and new nozzles; $p_{t,1} \approx 1500$ psia and z/L - 0.5 for 20-Inch Mach 6 CF_4 Tunnel.

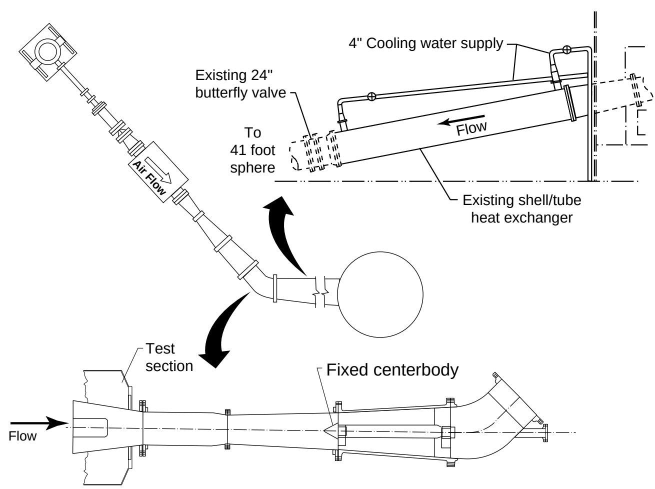


Fig. 16 Sketch of original diffuser and heat exchanger for 15-Inch HT Air Tunnel.

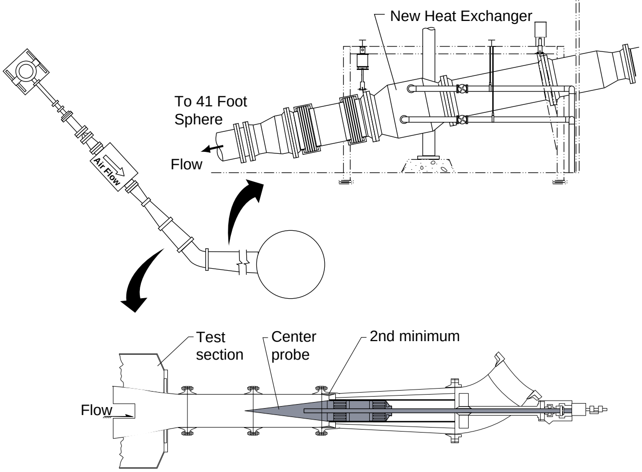


Fig. 17 Sketch of variable area diffuser (VAD) and new heat exchanger for 15-Inch HT Air Tunnel.

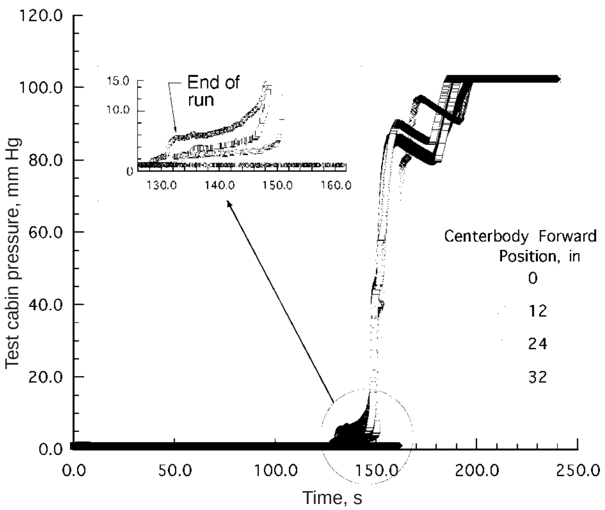


Fig. 18 Effect of area variation using translating plug (VAD) on total run time in 15-Inch HT Air Tunnel.

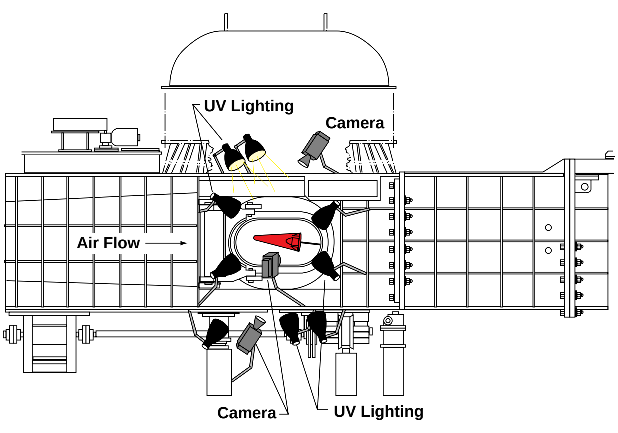


Fig. 19 Sketch illustrating optical access enhancements for 31-Inch Mach 10 Air Tunnel.

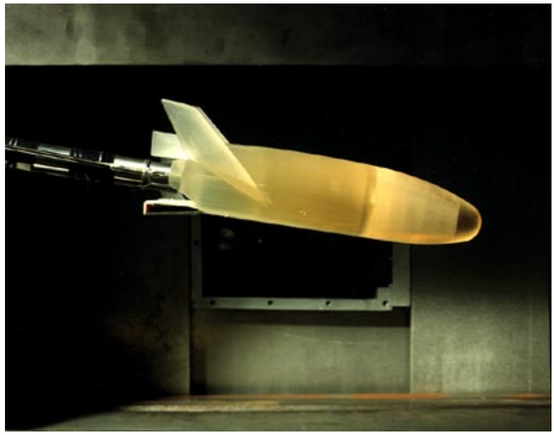
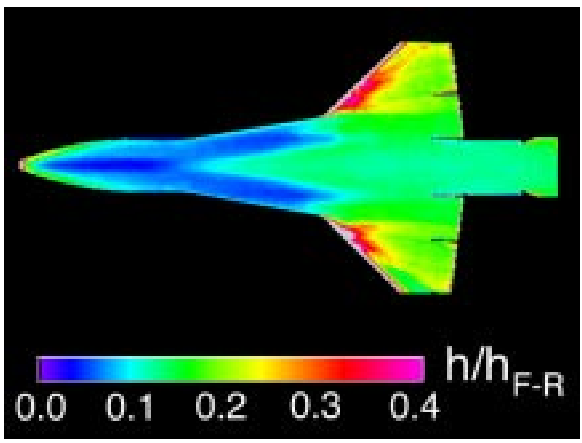


Fig. 20 Stereo-Lithography (SLA) X-33 model in 22-Inch Mach 15/20 Helium Tunnel.

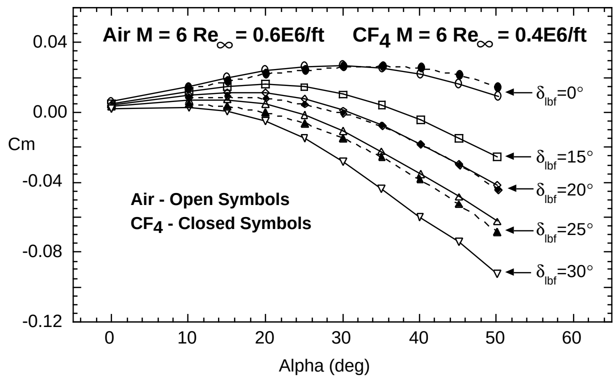
(a) Schlieren Photograph of X-33 Phase I concept in 20 Inch M6 CF_4.

Fig 21 Examples of typical aerodynamic and aeroheating data base development for X-33 phase I, X-34, and X-38 concepts.



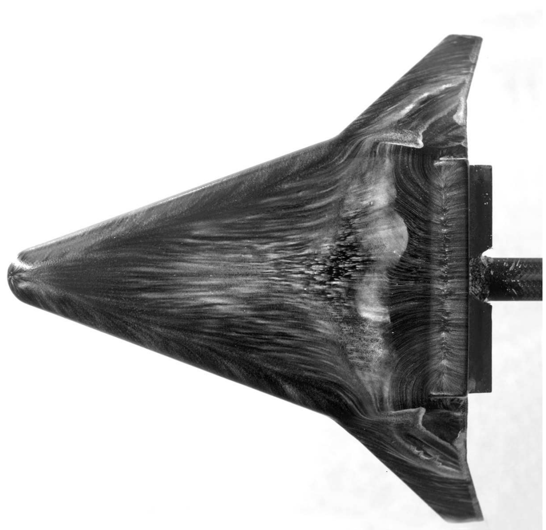
(b) Aeroheating - Thermographic Phosphor Technique; 20-Inch M6 air; α = 15°; X-34 concept.

Fig 21 Continued.



(c) Pitch characteristics at Mach 6 in CF_4 and air for X-38 concept.

Fig 21 Continued.



(d) Surface streamlines using oil flow technique for X-33 Phase I concept; 20-Inch Mach 6 Air; $Re_{\infty} \sim\sim 2.0 \times 10^{6}$/ft.

Fig 21 Concluded.